The AIRCRAFT ENGINEER
"FLIGHT" ENGINEERING SECTION
Edited by C. M. POULSEN
No. 118 (Volume XI No. 5) 10th Year November 28, 1935

COOLING PROBLEMS

With Particular Reference to the Work of the R.A.E. Tunnel

By G. P. DOUGLAS, M.C., D.Sc., A.F.R.Ae.S.

Résumé of Paper Read Before the Royal Aeronautical Society on November 18

COOLING problems have recently been occupying a considerable portion of the time of the Aerodynamics Department at the Royal Aircraft Establishment. The work had hitherto been of a theoretical nature combined with tests in the small tunnels. The new 24ft. tunnel has provided us with a very valuable piece of apparatus to deal with the problem. The work is proceeding and is still incomplete, but we are beginning to have a clear idea of the principles involved.

When we were preparing the scheme for this tunnel, no tunnel of the open jet type had been built in this country and very careful model experiments were made. First we built a model with working section one foot diameter and experimented until the flow was to our satisfaction. This one foot tunnel has since proved very useful and was used for the ice formation experiments described to you in the last lecture. This tunnel was then reproduced in its essential features to five times the scale and thoroughly tested. The resulting design fitted with a 500 h.p. motor gives us a very useful tunnel with a speed of 215 miles per hour. The 24ft. tunnel reproduces the 5ft. design to an increased scale.

The leading features are :—

Jet diameter	24ft.
Length of jet	44ft.
Collector dia. (maximum)	40ft.
Fan diameter..	30ft.
Length of centre line of return circuit	356ft.
Contraction ratio: (max. area/jet area)	3.53 : 1
Max. h.p. of fan motor	2,000 h.p.
Max. tunnel speed	115 m.p.h.

The building is of steel and ferro-concrete construction, and includes an aeroplane erecting shed to one side of the open jet (Fig. 1), and a tower above the jet containing a lift which carries the balances from which models are hung (Fig. 2), and containing the model rigging bays. This will allow models to be lowered into the jet, when there is a pause in full-scale work.

Since aeroplanes can be tested with their engines running, ventilation is provided, in order that the concentration of carbon monoxide in the exhaust gases shall not reach a dangerous proportion of the air circulating round the tunnel. Fresh air is let into the building through an annular space concentric with the nozzle. The lip of the collector is perforated, and air can be exhausted to the outside of the building by four extractor fans at a rate of 0.25×10^6 cu. ft. per minute. A wire mesh fan guard consisting of 16 gauge steel wire in 6-inch mesh was erected in the collector in front of the fan, to prevent objects from blowing into the collector and hitting the fan.

In the floor of the working section, a pit accommodates the main balance ; this is of weighbridge type, and measures lift up to 8,000lb. and drag from $\pm$ 4,000lb. simultaneously. The load can be balanced out by jockey weights to the nearest 100lb. and the remaining out of balance load can be read from an illuminated scale. The balance frame has three attachment points to take the feet of the pylon structure on which the aeroplanes are mounted before they are swung on to the balance by means of a jib crane.

Large scale models are slung from an overhead balance. Immediately above the working section there is an opening in the ceiling 20ft. by 18ft. 8in. through which the model, mounted on its balance car, can be lowered into the jet. The model bay above the working section is shown in Fig. 2.

The fan motor speed is controlled by a coarse and a fine control, the former operated by hand, and the latter by an automatic control device governed by the wind speed.

Tests on Complete Aeroplanes

The working section of the tunnel is large enough to accommodate an aeroplane of 50ft. span, but only the centre portion is in the air jet, and the tests are confined

to the effect of modifications to this centre portion. It is conceivable that the method might give misleading results owing to the limited jet. The change in the load distribution, due to a change in design, might be influenced by the jet boundary, or the change in design might modify the jet boundary itself.

plane 0.5 of the airscrew diameter behind the airscrew. No change could be detected.

Open jet tunnels are liable to pulsate. The actual phenomena are complicated, but the broad explanation is probably as follows: when an air jet is passing through still air the boundary is unstable, and vortex rings are

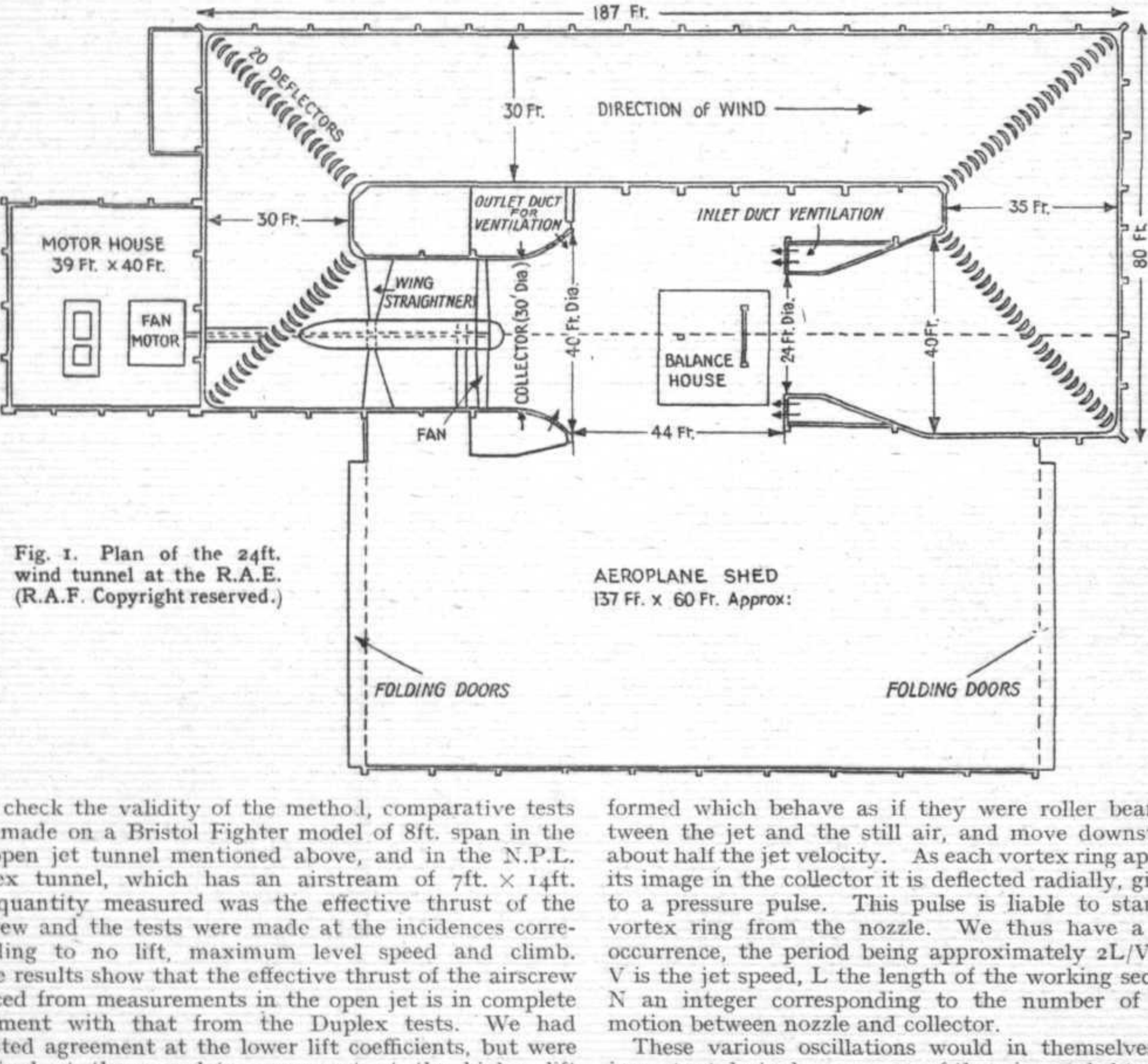


Fig. 1. Plan of the 24ft. wind tunnel at the R.A.E. (R.A.F. Copyright reserved.)

To check the validity of the method, comparative tests were made on a Bristol Fighter model of 8ft. span in the 5ft. open jet tunnel mentioned above, and in the N.P.L. Duplex tunnel, which has an airstream of 7ft. × 14ft. The quantity measured was the effective thrust of the airscrew and the tests were made at the incidences corresponding to no lift, maximum level speed and climb.

The results show that the effective thrust of the airscrew deduced from measurements in the open jet is in complete agreement with that from the Duplex tests. We had expected agreement at the lower lift coefficients, but were surprised at the complete agreement at the higher lift coefficients.

The influence of the airscrew, which was 0.4 of the jet diameter, on the jet boundary was also investigated in a

formed which behave as if they were roller bearings between the jet and the still air, and move downstream at about half the jet velocity. As each vortex ring approaches its image in the collector it is deflected radially, giving rise to a pressure pulse. This pulse is liable to start a new vortex ring from the nozzle. We thus have a periodic occurrence, the period being approximately 2L/VN, when V is the jet speed, L the length of the working section and N an integer corresponding to the number of rings in motion between nozzle and collector.

These various oscillations would in themselves be unimportant, but when any one of them is coupled to a resona-

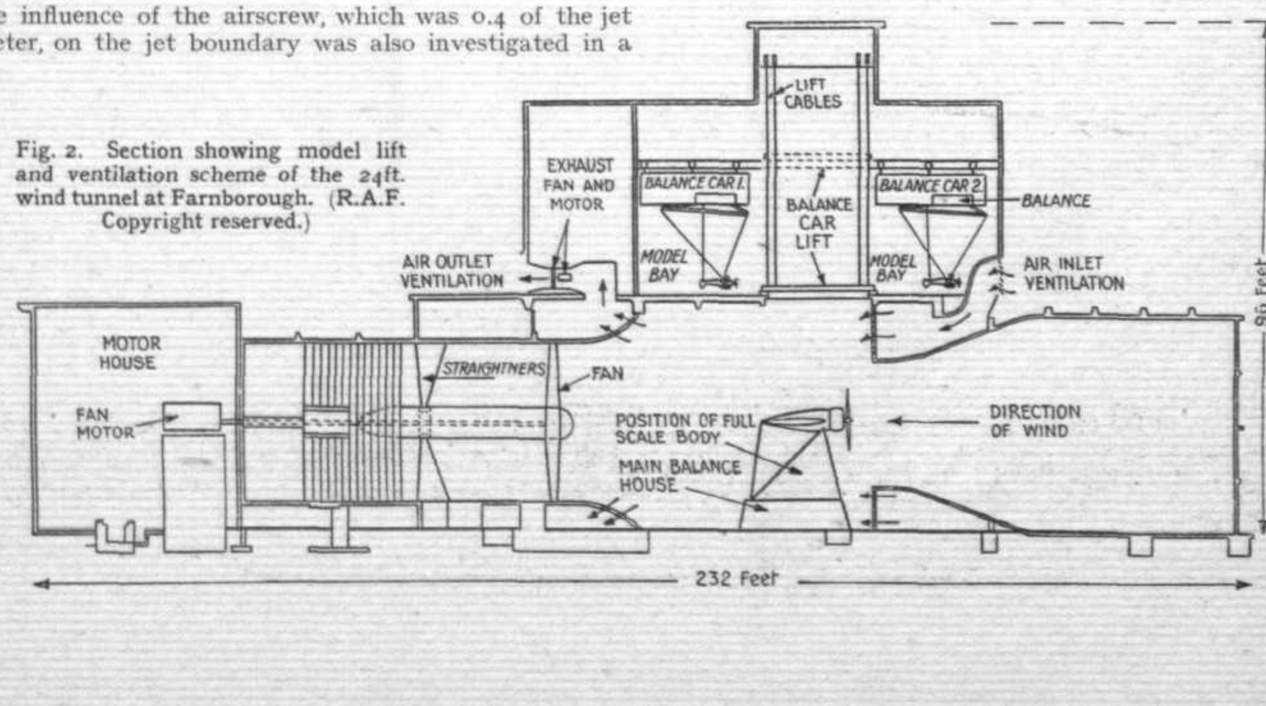


Fig. 2. Section showing model lift and ventilation scheme of the 24ft. wind tunnel at Farnborough. (R.A.F. Copyright reserved.)

tor of suitable frequency, oscillations of large amplitude may result. In a tunnel of the type we are considering, the return circuit can resonate as a tube with both ends open and it usually resonates with two nodes as indicated (Fig. 3). A cure can be effected either by introducing leaks near one of the nodes to damp the resonating properties of the return circuit, or by fitting tabs to break up the jet surface and prevent the formation of large vortex rings.

Serious pulsations occurred when the 24ft. tunnel was first run. The flow was observed by watching airspeed indicators connected to a point at the nozzle mouth in the centre of the jet, and also to the static pressure hole. At the centre of the jet, fluctuations were observed at 50 m.p.h.; at a mean speed of 80 m.p.h., 4 m.p.h. fluctuations were recorded, and 8 m.p.h. fluctuations at 100 m.p.h. The frequency of the fluctuations was 2.8 per second at 80 m.p.h. when an audible beat of this period was heard. This increased in intensity, and severe vibrations occurred in many parts of the building as the speed was raised.

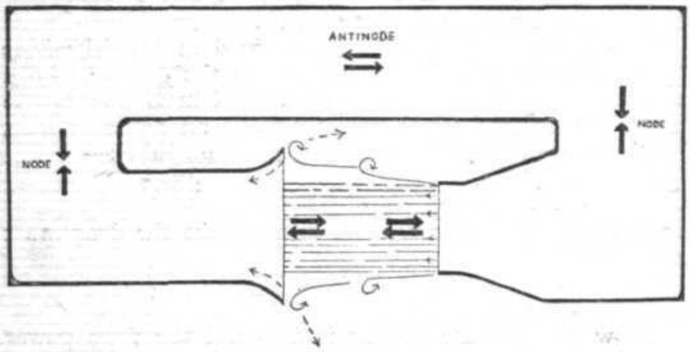


Fig. 3. Diagram illustrating pulsations in wind tunnel.

Fifteen tabs were then fitted in the mouth of the nozzle. A recording manometer was connected to the static pressure hole, and a continuous photographic record of pressure was made at various tunnel speeds, with the tabs projecting 3in., 6in., or 9in. into the jet. Without tabs, the fluctuation was equivalent to variations of 15 m.p.h. at a mean tunnel speed of 80 m.p.h. With tabs, part of the films showing the fluctuations are reproduced in Fig. 4, showing that 6-in. tabs give reasonably steady conditions. The audible beat and vibration of the building were stopped, and the variation of speed in the jet was less than 1 m.p.h. at all speeds. No difference could be detected in the power factor of the tunnel on adding tabs.

Fitting the tabs has affected the distribution of static pressure along the tunnel axis, since the nozzle was designed to make this constant without tabs. In consequence, the mean velocity in the central 20ft. of the tunnel increases as the distance from the nozzle increases. The velocity rises 1.3 per cent. in a length equal to one jet diameter. When time permits, tests of alternative methods of improving the steadiness of the flow will be considered.

The velocity distribution in any one section lies within $1\frac{1}{2}$ per cent. of the mean value over the central 20ft. diameter, and does not change appreciably as the speed is varied.

Critical Review of Design

The tunnel as built represents a very useful piece of apparatus. To keep down building costs, the return circuit was made as compact as possible, with a contraction ratio of $3\frac{1}{2}$ to 1. A slight increase would give more margin for errors in design. It is probably best to place the fan immediately behind the working section in an open working section tunnel. The uneven distribution after the open jet is immediately corrected, and the disturbance due to the model is partially evened out before the air goes round the first set of deflector aerofoils.

One question asked in connection with this tunnel is why is it not of the double return type? It is suggested that a single return tunnel must give poor velocity distribution because the air which has been following the long outer wall must have suffered a much greater loss in energy than the air which followed the short inner wall. Actually there is a secondary flow at each set of deflector aerofoils which transfers de-energised air from the outer to the inner wall and provides automatic compensation. Further, the double return is liable to produce a section of tired air in the middle of the test section; and it makes access to the working section difficult.

The second question is why was not the tunnel made bigger and why was the speed limited to 115 m.p.h. A tunnel to test even the smaller of the best service aircraft under maximum speed conditions would require twice the cross-section and at least three times the speed, so that instead of the present 2,000 h.p. for the fan, 100,000 h.p. would be required. Increase of size leads to reduction in sensitivity of balances and increased time on rigging and repair. The tunnel built represents a very convenient compromise for the type of work for which it is intended.

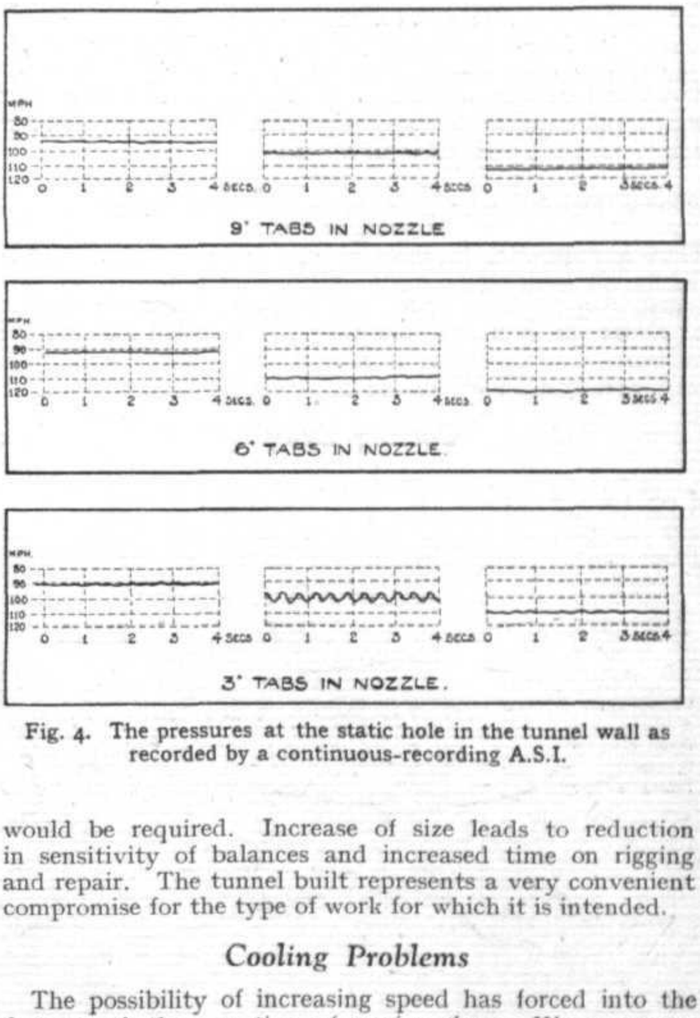


Fig. 4. The pressures at the static hole in the tunnel wall as recorded by a continuous-recording A.S.I.

Cooling Problems

The possibility of increasing speed has forced into the foreground the question of engine drag. We were considering recently the design of a commercial aeroplane suitable for the Atlantic passage, fitted with four 600 h.p. radial engines. If the drag per engine be taken at the very moderate figure of 15lb. at 100ft./sec., we find that at 265 miles per hour one of the engines is merely serving to neutralise the drag of the other three. If we could get rid of the engine drag the fourth engine would have been unnecessary, and all its weight and petrol would be available as pay load. If we consider still higher speeds and the 15lb. figure could not be reduced, we find that at 380 miles per hour the whole trust would be required to overcome the engine drag.

A few years ago, when the Schneider Trophy aircraft had to be cooled at very high speeds, the surface of the aircraft was used to dissipate the heat. The installation is heavy, and although there is a lack of experimental evidence we suspect that the heating of the aeroplane skin does increase its drag. The hot surface raises the temperature of the boundary layer and hence its viscosity,

so that a 100 deg. rise in temperature of the surface would increase the skin friction drag about 4 per cent. The power required to overcome a 4 per cent. increase in skin friction of the heated surface at 400 m.p.h. is comparable with the power expended in overcoming the total skin friction of a separate radiator having five times the cooling area and with a cooling velocity of one-fifth of the flight speed. The result, therefore, appears to be little better than could be achieved by an efficient system of low velocity duct cooling.

The importance of weight must be kept in mind. We can consider how much weight we would be justified in adding to achieve a given reduction in drag in an aircraft primarily designed for speed. Consider two aircraft of given loading (w) and power, having the same maximum speed in level flight (V). They will have the same thrust given by :—

$$T = k_{ow}\rho\,(W/w)\,V^2 + D_0\,(\rho/\rho_0)\,(V/100)^2$$

where W is the total weight, k_{ow} the drag coefficient of unit wing surface and D_0 the extra to wing drag at 100 f./s. The first term on the right-hand side is the wing drag and the second term the extra to wing drag. The only variables are W and D_0, and differentiating :—

$$dW/dD_0 = -\,w/k_{ow}\rho_0\,100^2.$$

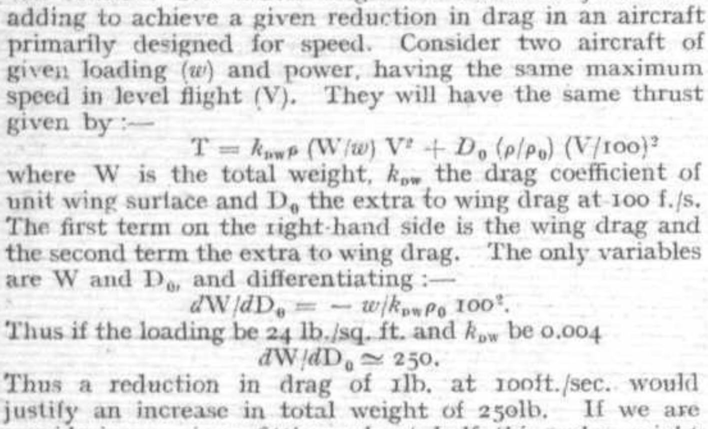
Thus if the loading be 24 lb./sq. ft. and k_{ow} be 0.004

$$dW/dD_0 \simeq 250.$$

Thus a reduction in drag of 1lb. at 100ft./sec. would justify an increase in total weight of 250lb. If we are considering a given fitting, about half this value might be taken, since in order to meet the specified conditions, the structure and wing weight of the aircraft would be increased by a similar amount to carry the fitting.

Cooling Drag

The net power required to cool an engine at low speeds is the product of the air flow through the cooling system and the pressure drop. For a liquid-cooled engine this power can be reduced indefinitely by making the radiator big enough, but using a radiator of reasonable size (say 40 calibres with an air speed 60 miles per hour in front of the radiator), the loss would be between 1 and 2 per cent. of the b.h.p. In the case of an air-cooled system the answer is less definite, but the loss is probably about 2 or 3 per cent. The value depends on the permissible cylinder temperature, and with the deep fins now employed an increase in cooling speed produces only a small increase in cooling, in fact to drop the temperature 25 deg. the cooling power must be doubled.



$\frac{h}{H}$	0·2	0·4	0·6	0·8
η	0·95	0·87	0·82	0·73

Fig. 5. Diagrams illustrating the theory of duct design.

If these low percentage cooling losses are to be maintained to high speeds, and the absurd losses of the previous paragraph avoided, we must have an efficient duct system in which the speed of the cooling air is reduced to that required for efficient cooling, and is discharged again without spoiling the flow round the aircraft.

The Theory of Duct Design (Fig. 5)

The theory of duct cooling has been developed independently by Capon and by Meredith, but I shall only mention the leading principles in this paper.

Consider any cooling system enclosed in a cowl whose "profile" drag is neglected. If H be the dynamic head of the outside stream and h the loss of head due to the engine or radiator which we regard as an actuator disc, then by Bernoulli's equation for any stream tube :—

$$p_0 + \tfrac{1}{2}\rho V^2 = h + p_0 + \tfrac{1}{2}\rho V_2^2$$
$$\text{or } H - h = \tfrac{1}{2}\rho V_2^2.$$

By considering the momentum in the wake the drag must be :—

$$D = \rho Q\,(V - V_2)$$

where Q is the rate of flow.

The work down in cooling is hQ, and the drag power expended is DV, so that the efficiency is given by :—

$$\eta = hQ/DV$$
$$= hQ/\{\rho QV\,(V - V_2)\}$$
$$= (h/2H)\,1/\{1 - \sqrt{(1 - h/H)}\}$$

The table shows the variation of η with h/H :—

h/H	0.2	0.4	0.6	0.8
η	0.95	0.87	0.82	0.73

This very important relationship, due to Meredith, enables us to compare the drag cost of any cooling system with that which is theoretically possible.

If for a given installation we regard the cooling pressure loss as constant, it appears that good efficiencies are possible over a wide range of flight speeds. At low values of h/H corresponding to very high speeds, efficiencies approaching unity are possible. The formula does not give a solution when the loss of head exceeds the dynamic head of the free stream, and we know that with trailing edge flaps a pressure 50 per cent. in excess of this can be obtained. The significance is probably that streamline flow is only possible when the internal pressure drop ~~is less~~ than the dynamic head, and any pressure in excess of this will be costly in drag.

The size of the exit for a given engine resistance and forward speed is obtained from the Bernoulli equation :—

$$p_0 + \tfrac{1}{2}\rho V^2 = h + p_1 + \tfrac{1}{2}\rho V_1^2$$
$$\text{and if } V_1 = Q/A$$
$$A^2 = \{H - (p_1 - p_0) - h\}/\tfrac{1}{2}\rho Q^2.$$

The term $p_1 - p_0$ is the increase in static pressure at exit over that at infinity. It may be taken to be determined by the external field, and it will generally be negligible.

If the issuing stream is being discharged near the front of the body, it is obvious that it should be discharged smoothly in the direction of the passing stream, to avoid spoiling the flow.*

It will be noted that the size of entry does not enter into the equations.

It has been pointed out that the essential power for cooling drag is small and it has been shown that the necessary flow can be induced at high speeds with high efficiency. We have, however, assumed that the air can be slowed without loss of energy or rather that the term "h," used for the engine loss, includes all losses in the ducts between entry and exit; if there is any considerable loss of head at entry and in slowing down the air, the actual efficiency may be much below that stated.

It is possible to slow the air without loss if it can be kept away from any surface, as happens with an ordinary pitot

* Actually the drag depends only on the mass and total head at exit for $D=\rho Q(V-v_2)$ and $v_2=\{\sqrt{(H-h)}\}/\rho$

tube. If, however, it is being slowed while passing over a surface, as soon as the boundary layer exceeds a certain thickness breakaway occurs, with a considerable loss of head. If the breakaway depends on the kinematic viscosity (v) velocity gradient and boundary layer thickness, the non-dimensional expression must be $(\delta^2/v)\,(dV/ds)$, where δ is the boundary layer thickness and s is the distance back from the nose,. and this must not exceed the critical numerical value. It is obvious that the air entry must be kept well forward where δ is small, and if for some reason the entry must be back, the air speed should be very gradually reduced, or some means taken to get rid of the thick boundary layer.

Tests in the 24ft. Wind Tunnel

The tunnel was formally opened in April, and apart from preliminary calibration tests it has been working continuously on cooling problems. The first problem was to determine the real magnitude of the engine and cooling drag of existing engines, a point on which there was considerable uncertainty, and the second was to see how far these items could be reduced by the application of the principles just considered.

Two installations were chosen for our preliminary investigations :—

(1) The Gauntlet, fitted with a radial air-cooled Mercury VI.S. engine having a simple ring cowl, and

(2) The Bulldog, fitted with a Rapier air-cooled in-line engine.

The Gauntlet is typical of the best modern practice of a fixed cowl unbaffled installation, while the Rapier Bulldog, on the other hand, is a completely baffled installation which lends itself to experiments on controlled cooling flow.

In both series of tests, measurements were made to determine :—

(*a*) The quantity of air passing through the cowl.

(*b*) The cooling drag.

This was deduced by difference from measurements of the drag of the aircraft with engine, and the drag of the aircraft with the engine removed and replaced by a well-shaped nose fairing. The drag difference found by these tests was taken to be the cooling drag of the installation. The measurements, of course, could only be made without slipstream.

(*c*) The drag loss inside the cowl.

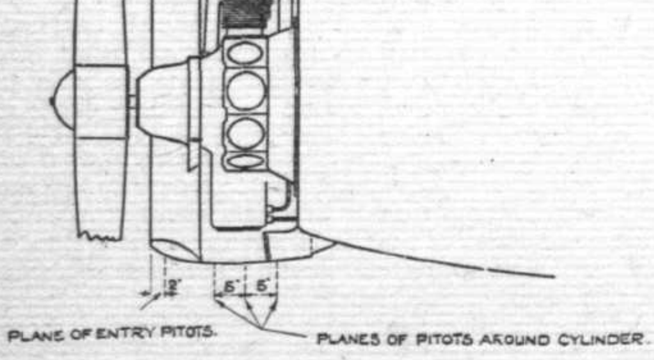


Fig. 6. Diagram of the installation of the Mercury VI.S. and engine cowl on Gloster Gauntlet.

This was obtained from measurements of the total head and velocity of the air leaving the cowl. The method is analogous to the determination of the drag of a wing from the pressure and momentum in the wake. The measurements were made with and without slipstream.

The difference between (*b*) and (*c*) gives the spoiling effect on the engine and cowl installation.

I shall describe first the Gauntlet tests. The cowl is shown in Fig. 6. In these tests the total head and velocity over the inlet and outlet areas of the cowl were determined from readings of combined pitot static tubes. Three such tubes were used to explore the cowl entry and four tubes were fixed in the exit. These tubes were spaced at various angular positions around the cowl and were moved in turn to four positions radially across both the entry and exit of the cowl.

The tests were made with and without airscrew running, and in the tests without airscrew the airscrew was removed and a small gear case fairing was fitted.

The tests with airscrew were made at a series of speeds and at engine revolutions corresponding to level flight and climb advances respectively.

In addition to the flow measurements the drag of the Gauntlet was measured and the cooling drag was determined by repeating the drag experiments with the engine and cowl replaced by a nose fairing to the fuselage.

The main results are given in the table below. The cooling losses are given in h.p. and as percentages of the engine power.

	Level flight at 15,000ft. at 230 m.p.h.		Climb at 13,000ft. at 139 m.p.h.	
	With airscrew.	No airscrew.	With airscrew.	No airscrew.
Engine b.h.p. ..	675	—	640	—
Flow through cowl (lbs./sec.)	33.2	38.2	20.4	—
Flow through cowl (lbs./sec./h.p.) ..	0.049	—	0.032	—
H.P. absorbed in cowl deduced from flow measurement ..	34 (5%)	19.5 (3%)	6.5 (1%)	—
Cooling drag from balance readings (lbs. at 100 f.p.s.)	—	12.4	—	—
H.P.	—	54 (8%)	—	—

Inspection of the table shows that under level flight conditions :—

1. With no airscrew the cooling drag is 8 per cent. of the b.h.p. ; 3 per cent. of the b.h.p. is absorbed in the cowl, so the spoiling drag is about 5 per cent.

2. The airscrew increases the power absorbed in the cowl from 3 to 5 per cent. of the engine power and seriously reduces the flow. This is due to the breakaway of the flow behind the boss. Under flight conditions the total cooling drag must thus be at least 10 per cent., *i.e.*, 2 per cent. above that deduced from the tests without airscrew.

3. If the engine is adequately cooled on the climb when the power absorbed is only 1 per cent. of the b.h.p., the flow at top speed must be greatly in excess of what is required. An excess of 50 per cent is computed.

In any air-cooled engine installation the cowl must provide sufficient cooling air to satisfy the conditions experienced on the climb. These conditions are always more severe than those met with at top and cruising speeds, and consequently the cowl that satisfies the climb conditions invariably permits more cooling air to pass through the cowl than is required to cool the engine adequately at the higher speeds.

We have just seen in the case of the Gauntlet cowl, that a mass flow approximately 50 per cent. in excess of the amount required for cooling the engine was passing through the cowl under level flight conditions. By suitable throttling of the emergent stream from the cowl, therefore, it should be possible to decrease the power losses inside the cowl and also to increase the efficiency of the induction of the flow by a relative increase of the velocity of its wake relative to the aeroplane.

To investigate the possible reduction of drag resulting from the application of this principle, some controlled

cowling tests were carried out on the Rapier Bulldog and these tests will now be described.

Tests on Rapier Bulldog

The first series of tests made on the Rapier Bulldog was with a cowl arrangement that had been flown full scale. These tests were made to establish the cooling drag and also to furnish a basis of cooling flow on which the subsequent controlled cooling experiments could be considered; the flight tests with this particular cowl arrangement having shown that the cooling flow was adequate. The method of tests followed closely on the lines of the tests already outlined in the case of the Gauntlet. Details of the cowl are shown in Fig. 7.

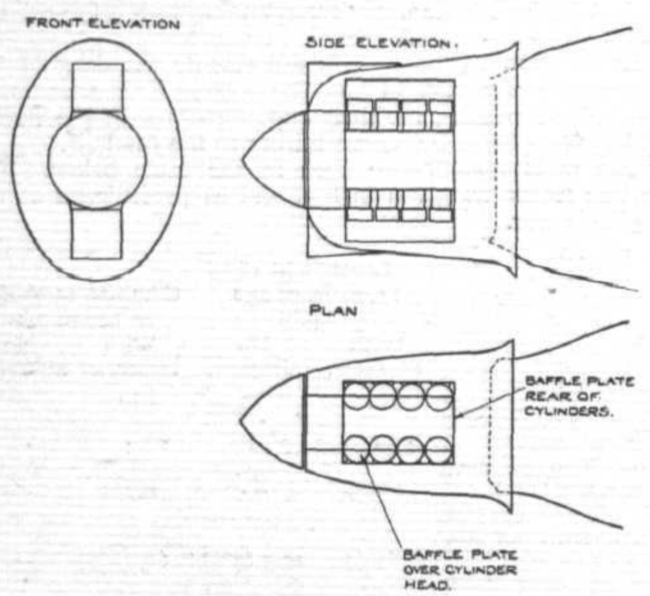


Fig. 7. Diagrammatic sketch showing internal baffle system and cowl as originally fitted for cooling tests on Rapier-engined Bulldog.

Flight tests had shown that the engine was adequately cooled on the climb at 95 miles per hour and from the flow measurements it was deduced that 0.32 cu. ft. (0.024lb.) of air per second per horse-power was required to cool the engine at maximum power.

The cooling drag at level flight attitude was 12.2lb. at 100 feet per second corresponding to 6.2 per cent. of the b.h.p. at the top level speed of 150 m.p.h. and the flow through the cowl was 28 per cent. in excess of the actual quantity required to cool the engine.

Passing now to the controlled cowling tests, the cowl was redesigned, first by incorporating controlled exits, and secondly by adding fairing pieces to the entries (Fig. 8). The points in the new design are that the flaps allow the exit air to discharge without undue disturbance to the outside flow. The entry has been carefully shaped to prevent breakaway for the flow over the range of forward speed required. Wool tuft exploration was found of great assistance in developing the entry shape. With the flaps set to give adequate cooling the level flight cooling drag is now 3.0 lb. at 100 feet per second, *i.e.*, at 1.5 per cent. of the b.h.p. of the engine. The corresponding figure deduced from flow measurements for the power absorbed in the cowl was 2½ per cent., which seems to indicate a negative spoiling drag. The nose fairing used to obtain the zero for the drag tests may not have been perfect, but a considerable portion of the resistance of the aeroplane is due to the body and the projection behind the engine and their drag may have been less in the de-energised air.

The effect of opening the flaps on the flow and drag may be seen from the following table and the large increase in drag when the flaps are opened to large angles will be noted.

TABLE SHOWING COOLING DRAG AND QUANTITY OF AIR PASSING THROUGH THE COWL.

Cowl detail.	Drag due to cooling lb. at 100 ft. per sec. Level attitude.	Climb attitude.	Quantity of air passing through cowl in cu. ft. at 100ft. per sec. Level attitude.	Climb attitude.
Cowl No. 1 (as flown)	12.2	10.8	72	68
Controlled cooling cowl with faired entry scoops inlet area 0.91ft.—				
Exit area:				
1.91ft.^{2}	16.4	17.2	69	—
1.06ft.^{2}	4.5	6.7	59	—
0.55ft.^{2}	2.5	4.5	52	—
0.23ft.^{2}	2.4	—	48	—

Tests were made also to determine the drags of the external exhaust and external oil-cooling systems fitted to the Rapier Bulldog. These tests showed that the drag due to the exhaust pipes was about 5lb. at 100 feet per sec. and of the external oil-cooler 1.7lb. at 100 feet per sec. Both these losses should be eliminated on a high speed aircraft.

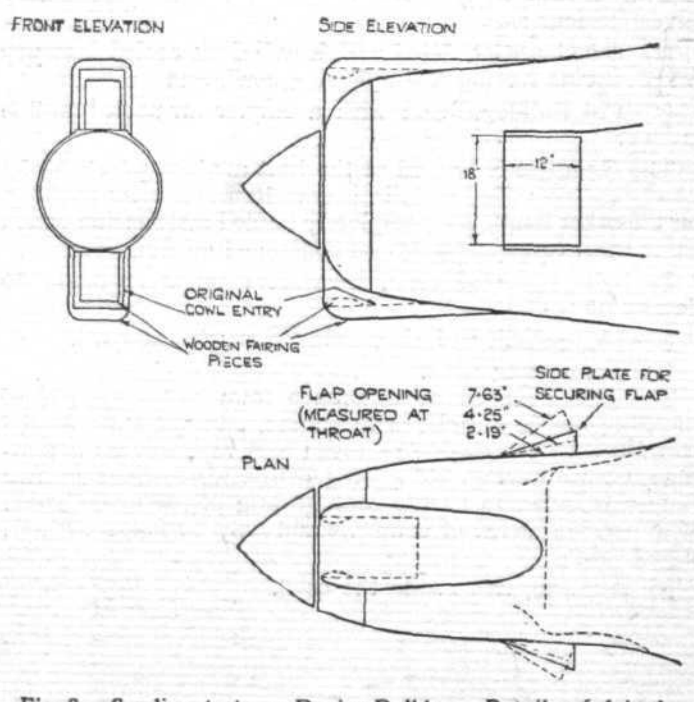


Fig. 8. Cowling tests on Rapier-Bulldog. Details of faired entry used in conjunction with controlled exit.

Reviewing the cost of cooling results for the Rapier Bulldog and Gauntlet installations it appears that at maximum level speed, using fixed cowls, the cooling drag of the Rapier installation is costing about 6 per cent. of the b.h.p., while for the Gauntlet the cost is about 10 per cent. It has been shown from the Rapier that by using a cowl with adjustable exit this loss can be reduced by about 5 per cent.; for the Gauntlet a reduction of about 9 per cent. is estimated. The actual minimum drag cost is a little arbitrary as it depends on the nose shape relative to which the drag differences have been measured, but is clearly very low. The Gauntlet result is interesting in showing that this low cooling drag is possible without the addition of an internal baffling to the cylinders.

Exit Conditions

The conditions to be satisfied at exit are pretty clear apart from the possibility of the "tired" air reacting on the main flow. It is obviously very bad practice to discharge de-energised air at any point where the flow has small stability, *e.g.*, on the upper surface of a wing or

above a wing root fillet. Release on the wing lower surface is relatively harmless, but probably discharge at the trailing edge is ideal. Apart from this spoiling there is no mystery about the discharge. The cooling air must have a smooth easy passage to join the outside stream. An engine installation under given power and flight conditions requires to maintain a certain pressure across it for cooling flow. For the Rapier at full power this head appeared to be about 8.3 inches of water. If the flight range extends to sufficiently low speeds we may have to use flaps set to large angles to induce the cooling flow. If, however, the climb speed is high enough so that a pressure difference less than the dynamic head will supply this head there is no need to use high drag flaps and the exit should be designed to give the maximum momentum to the issuing stream.

Recovery of Heat Energy

The increasing dynamic compression which is available as flight speeds increase make it possible to recover energy from the heat added to the cooling flow. The heat is added to the air after compression and gives increased momentum to the discharging stream. The efficiency of conversion is $1 - (p_0/p)(\gamma - 1)/\gamma$ where p is the absolute pressure at which the heat is added and p_0 is the static pressure of the outside stream. The value of this, if the full adiabatic rise were obtainable, is 4 per cent. at 300 m.p.h., and it increases as the square of the forward speed. The cooling power is of the order of half the engine power, and if the exhaust heat be also available at least twice the engine power is available. Mr. Meredith has shown that for a typical radiator in a duct the total drag could be reduced to zero at 300 m.p.h. by making use of the cooling energy or at 140 making use of the cooling and exhaust energy; above these speeds a thrust is obtained. Similar results could be given from an air-cooled installation. Preliminary experiments have confirmed that a reduction in drag approaching the order of that calculated can be obtained.

Concluding Review

I have indicated the lines on which we must proceed to keep down cooling losses as flight speeds increase. The dangers of serious entry losses must be kept in mind.

Oil-coolers can be of the internal type, and when provided with a duct so that the cooling air discharges at the trailing edge we have found the drag too small to be measured.

In many cases baffles can be used to reduce cooling losses, but the Gauntlet results I have described suggest that efficient cooling at low power cost can be obtained without that complication.

Increase of speed provides us with a convenient adiabatic compression, and the possibility of using this heat cycle so that our cooling will add to the propulsive instead of to the drag force.

STALLING of TAPERED WINGS

By P. P. NAZIR, A.F.R.Ae.S., M.Inst.B.E.

Résumé of Paper Read Before Students' Section of the R.Ae.S. on November 19, 1935.

AFTER pointing out that for many years it has been known that a rectangular wing stalls first in the centre, and burbling then spreading towards the tips, Mr. Nazir related results of an investigation made at Queen Mary College on the three wings shown in Fig. 1. Filaments of swan's down were fixed in tiny holes in the upper surface of the wings to form streamers for visual observation of the flow. The rectangular wing was of R.A.F. 38 section, the 1 in 5 straight taper wing of the same section, and the moderately tapered wing (1 in 2) had a different section. The following table shows the percentages of wing area stalled at different angles for the three wings.

	INCIDENCE.									
Type of Wing	2°	4°	8°	11°	12°	13°	14°	16°	17°	18°
Rectangular: Per cent. Wing Stall	7	7	33	47	60	60	73	87	87	100
Tapered: Per cent. Wing Stall	13	27	40	67	80	93	93	93	100	100
Special: Per cent. Wing Stall	13	13	40	73	73	80	80	100	100	100

The distribution of stall over the wings is indicated in Fig. 2. It was found that all streamers maintained a fairly steady position at 0 deg. incidence. Streamers 0 and 1 on the rectangular wing gradually became unsteady as incidence increased. The unsteady motion was considerably increased at 9 deg. and creeping towards the tips, involving Nos. 0, 1, 2 and 3 on each side. When incidence was increased to 12 deg., No. 4 streamer on each side joined in. At 14 deg. No. 5 streamer was involved, and at 16 deg. No. 6 streamer.

Mr. Nazir, who is a Government of India Special Research Scholar at Queen Mary College, where he is developing an invention on lateral control, can claim the honour of being the first Parsee to read a technical paper on an aerodynamic subject before the R.Ae.S.

A marked difference was observed in the case of the tapered wing. No. 7 streamer on each side was the first to become unsteady. With increase in angle of incidence the burbling crept *inwards*, contrary to the rectangular wing. At 9 deg. incidence, streamers No. 7, 6 and 5 on each side became affected, and at 11 deg. Nos. 4 and 3 joined in. All streamers were highly disturbed at 13 deg. except No. 0.

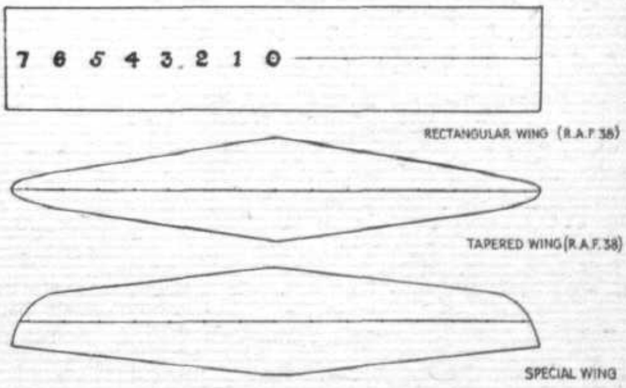


Fig. 1. Plan forms of three wings tested.

For the moderately tapered wing the growth of the disturbance was different. No. 5 streamer on each side first became unsteady, and later burbling spread both ways, *i.e.*, outwards towards the tip and inwards towards the centre. At 11 deg. streamers Nos. 7, 6, 5, 4 and 3 joined in, and No. 2 was affected at 13 deg. On the rectangular wing the streamers reversed their direction at 18 deg., but No. 7, the tip streamers, did not appear very disturbed, nor did they reverse their direction through an incidence range of 75 deg.

The last tip streamers on the tapered wing were unsteady

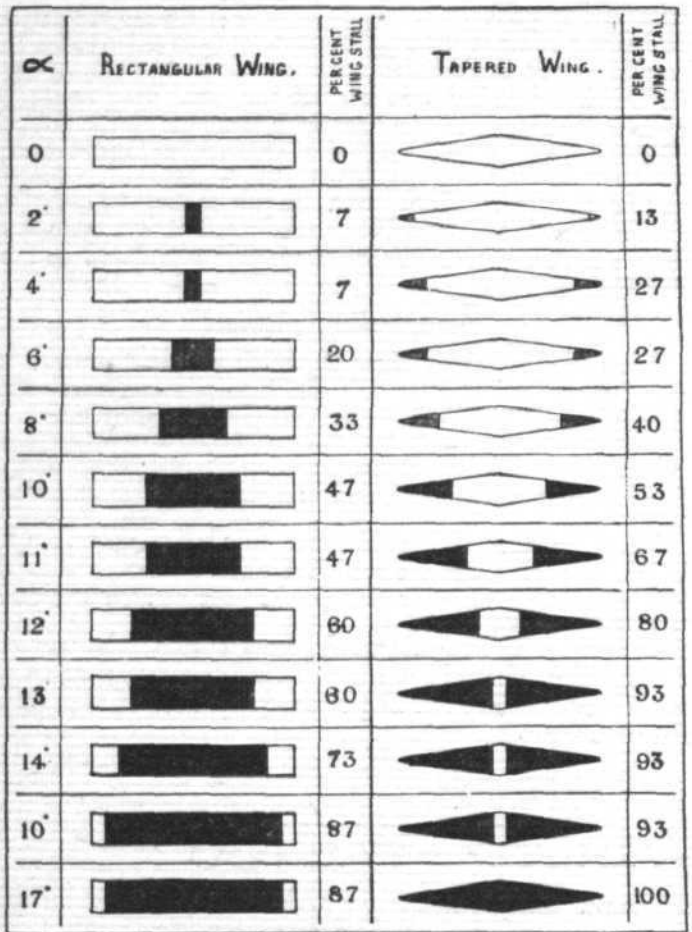


Fig. 2 Distribution of stall (black areas) on rectangular and tapered wings.

almost from the beginning, and pointed towards the centre of the wing even at small incidences.

At the angle of maximum lift, just before the stall, interesting contrasts between different wings are shown (Fig. 3). The tapered wing clearly shows severe disturbances at the tips, but on the rectangular wing there is no reversal of streamer directions. On the moderately tapered wing the reversal happens intermediately between tips and centre.

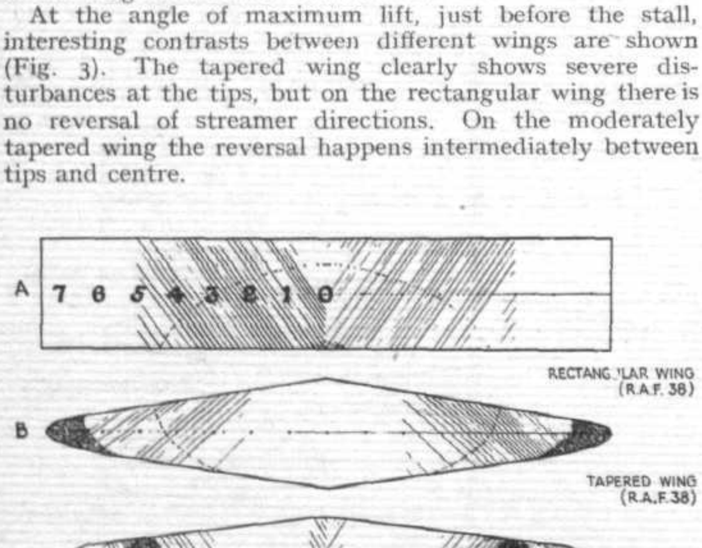


Fig. 3. Distribution of disturbances on tapered wings at angle of maximum lift. There is no such disturbance on the rectangular wing.

An attempt was made to estimate the strength of the disturbance by recording complete reversals of the streamers. On the rectangular wing the first reversal occurred at 15 deg. incidence, when 9 streamers reversed almost simultaneously. Eleven reversed completely at 18 deg. In the case of the tapered wing the two tip streamers reversed at 11 deg., then 8 streamers at 14 deg. and 14 streamers at 18 deg., *i.e.*, all except the central one. On the moderately tapered wing the first reversal occurred at 11.5 deg. incidence, when No. 5 on each side completely reversed. At 14 deg. 12 reversed, and at 18 deg. all 15 reversed. The experiment (Fig. 4) shows that at low

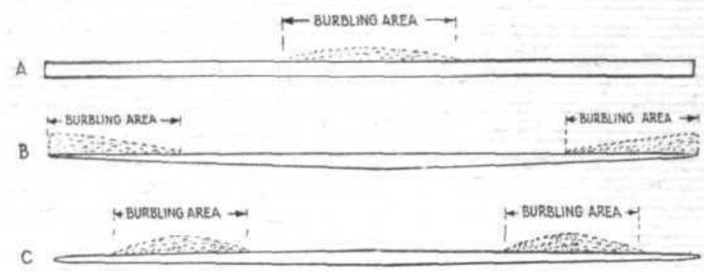


Fig. 4. Three different types of stall. A is the rectangular wing, B the sharply tapered, and C the moderately tapered.

Reynolds Number, three different types of stall occur on wings of different taper. The rectangular wing is subject to central stall, the moderately tapered to intermediate stall, and the heavily tapered to tip stall.

On the rectangular wing the central stall does not interfere with the control area until an incidence of 14 deg. has been reached. The tip stall affects control from the very beginning. The whole control area is rapidly occupied with an increase of 6 deg. incidence only, and therefore there is a possibility of inefficient control during normal

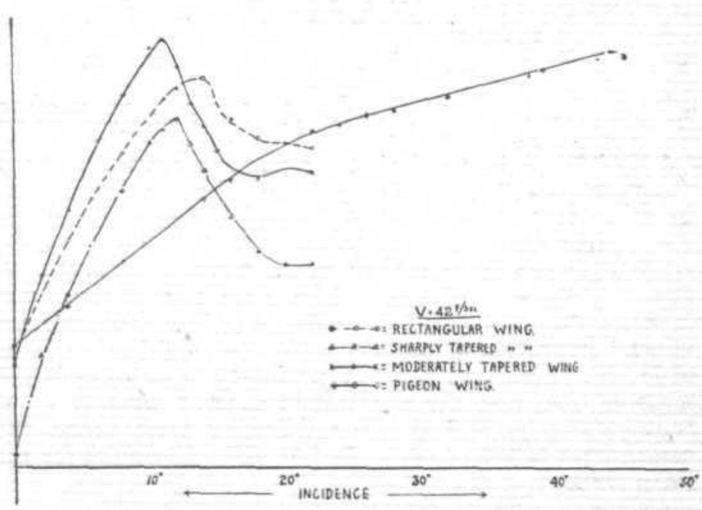


Fig. 5. Comparative lift curves. That of the pigeon's wing is interesting.

manœuvres such as take-off and climb. The intermediate stall partly occupies the control area of 4 deg. and with an increase to 10 deg. the whole control area is affected.

Mr. Nazir then showed slides from a film illustrating burbling on these three wings, and also on a pigeon's wing. The latter showed central stall, similar to that of a rectangular wing.

Fig. 5 shows lift curves of the three wings. The maximum lift of the rectangular wing occurs at 14 deg., after which the lift falls off in the normal manner. The lift of the tapered wing reaches a maximum at 12 deg. incidence, with a correspondingly steeper drop in lift. The moderately tapered wing reaches its maximum lift at 11 deg. incidence, and the lift then falls off rather steeply.

In conclusion, the lecturer suggested that it might be worth while to analyse the different flows that occur after the stall, and that a comprehensive study would give the best form of tapered wing, which might possibly be designed to stall first in the centre. He also suggested that his results might be compared with full scale by studying streamer movements in flight.